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NAVIGATOR STUDY OF ELECTRIC PROPULSION FOR UNMANNED SCIENTIFIC MISSIONS

CONSTANT POWER MISSION ANALYSIS

by H. Brown and J. R. Taylor

Prepared by
GENERAL ELECTRIC
Philadelphia, Pa.
for Lewis Research Center

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1.0 INTRODUCTION

This topical report presents the results of studies performed by the General Electric Missile and Space Division during an additional two-month extension of Contract NAS 3-2533 for the NASA Lewis Research Center. Five reports (References 1-5) were issued during the original contract under the title Research on Spacecraft and Powerplant Integration Problems. Three additional reports (References 6-8) were issued at the completion of the original contract extension under the title NAVIGATOR, Study of Electric Propulsion for Unmanned Scientific Missions.

The primary objective of the program has been the identification of nuclear-electric power generation system requirements for performing unmanned scientific exploration missions throughout the solar system beyond the capabilities of presently envisioned chemical and nuclear rocket propelled vehicles. This class of missions is referred to in these reports as NAVIGATOR missions. Both the original contract and the first contract extension have been structured to determine optimum power requirements for maximizing payload capabilities at constant trip time for each individual mission. These two study efforts differed in that the original contract considered post-SNAP-50 nuclear powerplant technology with electric propulsion initiation from earth orbit, and that the original extension considered an earlier powerplant technology with chemical propulsion substantially beyond earth escape in order to reduce trip time requirements. These results have generally indicated a different power requirement for each mission.

The present study differs from the previous efforts in that specific power levels of 100, 160, 240, and 320 kw have been selected and optimum mission performance determined for each mission at these discrete power levels. Two levels of powerplant specific weight have been considered corresponding to the early and improved technology levels of Reference 7. This report summarizes the results of these studies.

2.0 SUMMARY

The present investigation considered single burn low thrust orbiter missions to all of the planets in the solar system as well as fly-by missions to the four major outer planets. The NAVIGATOR vehicle was assumed to be boosted beyond earth escape by a multistage Saturn V booster.

The fly-by missions were restricted to a propulsion time of 10, 000 hours. In order to deliver the minimum payload considered to be of useful scientific value, the Saturn fly-bys required a total trip time of only 11, 300 hours with early technology, and the Uranus missions required about 22, 000 hours. There was a small decrease in trip time with increasing power. By improving the level of technology, the Saturn trip time was reduced to values of 10, 000 hours and less, indicating that the full 10, 000 hours of propulsion time was not required. The trip time for the Uranus missions was about 18, 500 hours. The minimum useful payload for the Neptune and Pluto fly-bys could not be delivered in less than 25, 000 hours without increasing the power above 320 kw or increasing the propulsion time above 10, 000 hours.

Propulsion time was allowed to vary for the major planet orbiters in order to optimize the Saturn V booster requirements. The Jupiter I and Jupiter II missions required about 18, 500 hours and 26, 000 hours trip time, respectively, in order to obtain the minimum payload with early technology. Propulsion times for these missions ranged from 6, 000 to 11, 000 hours and from 13, 000 to 19, 000 hours. With improved technology, the Jupiter I and Jupiter II trip times were about 16, 000 and 21, 500 hours with propulsion times from 5, 000 to 10, 000 hours and 9, 000 to 16, 000 hours, respectively. Increasing the power improved performance for both of these missions. The Saturn I orbiter required a trip time of 36, 000 hours with early technology and 29, 000 hours with improved technology. Propulsion times ranged from 8, 000 to 12, 000 hours and from 6, 000 to 10, 500 hours. For this mission, with early technology, the lower power levels appeared to yield slightly lower trip times although propulsion times were higher. Analysis of the remaining major planet orbiters showed that they required trip times in excess of 40, 000 hours.

It was found that the Venus and Mars orbiters did not require a booster as large as the assumed Saturn V and, in fact, the NAVIGATOR powerplant was underpowered for this booster. Acceptable performance for these missions could be obtained by off-loading the Saturn V or by using a smaller booster. The Saturn V, on the other hand, seemed suitable for the Mercury orbiter at the higher power levels. Trip times of about 4,200 hours with propulsion times of 3,400 hours were obtained with early technology. Corresponding values with improved technology were 3,400 hours and 2,900 hours.

3.0 MISSION SPECTRUM

Planetary fly-by and orbiter missions have been investigated in which a single continuous nuclear-electric propulsion period is utilized. This propulsion period occurs immediately after chemical or nuclear rocket burnout in the case of the fly-by missions and after the heliocentric coast in the case of the orbiters. The scope of the study has, however, been limited to include only the Saturn, Uranus, Neptune, and Pluto fly-by missions and the orbiter missions considered in Reference 6. Table 3-1 summarizes the individual missions considered.

TABLE 3-1. NAVIGATOR MISSION SUMMARY

Mission Type	Mission	Terminal Conditions
Fly-by	Saturn	Optimum Fly-by
	Uranus	1975-Fly-by
	Neptune	1986-Fly-by
	Pluto	1986-Fly-by
Orbiter	Mercury	2000 Mile Radius
	Venus	5000 Mile Radius
	Mars	3000 Mile Radius
	Jupiter I	1, 170, 000 Mile Radius
	Jupiter II	262, 000 Mile Radius
	Saturn I	760, 000 Mile Radius
	Saturn II	44, 000 Mile Radius
	Uranus	20, 000 Mile Radius
	Neptune	20, 000 Mile Radius
	Pluto	5, 000 Mile Radius

The factors considered in selecting the terminal conditions as related to the mission objectives are discussed in References 4 and 6.

4.0 PROPULSION REQUIREMENTS

The propulsion requirements for each of the NAVIGATOR fly-by and orbiter missions have been obtained from the one-dimensional correlation technique of Reference 6. This section describes the essential features of this technique and indicates the resulting propulsion requirements used for each mission. Reference 6 should be used for a more detailed description of the process.

The heliocentric acceleration-time history of an optimum power-limited (variable specific impulse) trajectory is assumed to be approximated by the following linear relationships:

$$\text{Fly-by} \quad - a = a_o \left[1 - (t/t_h)\right] \tag{4.1}$$

$$\text{Orbiter} \quad - a = a_o \left[1 - 2\,(t/t_h)\right] \tag{4.2}$$

The one-dimensional characteristic length is then defined by the equation:

$$L = \int_o^{t_h} \int_o^{t_h} a \, dt^2 \tag{4.3}$$

and the corresponding propulsion function by:

$$J = \int_o^{t_h} a^2 dt \tag{4.4}$$

Equations (4.1) through (4.4) can then be combined to obtain expressions for characteristic length as a function of trip time and the propulsion function:

$$\text{Fly-by} - L = \sqrt{\frac{J t_h^{\,3}}{3}} \tag{4.5}$$

$$\text{Orbiter - } L = \sqrt{\frac{J t_h^3}{12}} \qquad (4.6)$$

For the fly-by missions, the heliocentric time is the same as the total trip time.

The resulting equations (4.5) and (4.6) were then used in conjunction with the results of a series of optimum power-limited trajectory calculations obtained from the calculus of variations approach of Reference 9. The results of this process indicated that the characteristic length for each fly-by and orbiter mission could be represented by an empirical function of trip time of the form:

$$L = L_o - \left[\frac{2(L_o - L_m)}{t_m}\right] t_h + \left[\frac{L_o - L_m}{t_m^2}\right] t_h^2 \qquad (4.7)$$

Table 4-1 summarizes the numerical values of the empirical constants of equation (4.7) for each of the NAVIGATOR missions investigated. It has been shown in Reference 6 that these data can be used to define propulsion requirements for optimum coast-constant specific impulse trajectories.

TABLE 4-1. NAVIGATOR MISSION REQUIREMENTS

Mission	Type	t_m-hrs.	L_m-$(10)^6$ miles	L_o-$(10)^6$ miles
Saturn	Fly-by	9,946	589.6	679.2
Uranus		14,965	1,350	1,541
Neptune		17,454	2,313	2,494
Pluto		17,454	2,313	2,494
Mercury	Orbiter	2,381	39.00	57.23
Venus		3,452	18.49	29.00
Mars		6,000	27.68	46.70
Jupiter		14,086	306.6	383.5
Saturn		17,544	644.8	750.1
Uranus		25,747	1,460	1,606
Neptune		36,184	2,519	2,667
Pluto		36,184	2,519	2,667

Figure 4.1 illustrates the one-dimensional velocity diagram for both the fly-by and orbiter missions. These diagrams have been used as a guide in the development of a set of comparable characteristic length equations for the constant specific impulse mode of operation. The constant specific impulse (and constant thrust) acceleration equations

$$a = \frac{a_o}{1 - \frac{a_o t}{V_j}} \qquad \text{(during propulsion)} \qquad (4.8)$$

$$a = o \qquad \text{(during coast)} \qquad (4.9)$$

are substituted into equation (4.3) to obtain the following:

$$\text{Fly-by} - L = V_o t_h + V_j \left[t_{ph} + (V_j/a_o - t_h)\right] \ln \mu \qquad (4.10)$$

$$\text{Orbiter} - L = V_o (t_h - t_{ph}) + V_j (V_j/a_o - t_{ph}) \qquad (4.11)$$

The resulting equations (4.7), (4.10), and (4.11) have been used in conjunction with conventional system evaluation relations to obtain the heliocentric propulsion time requirements as a function of mission trip time and system jet velocity (specific impulse), initial acceleration, and initial hyperbolic excess velocity.

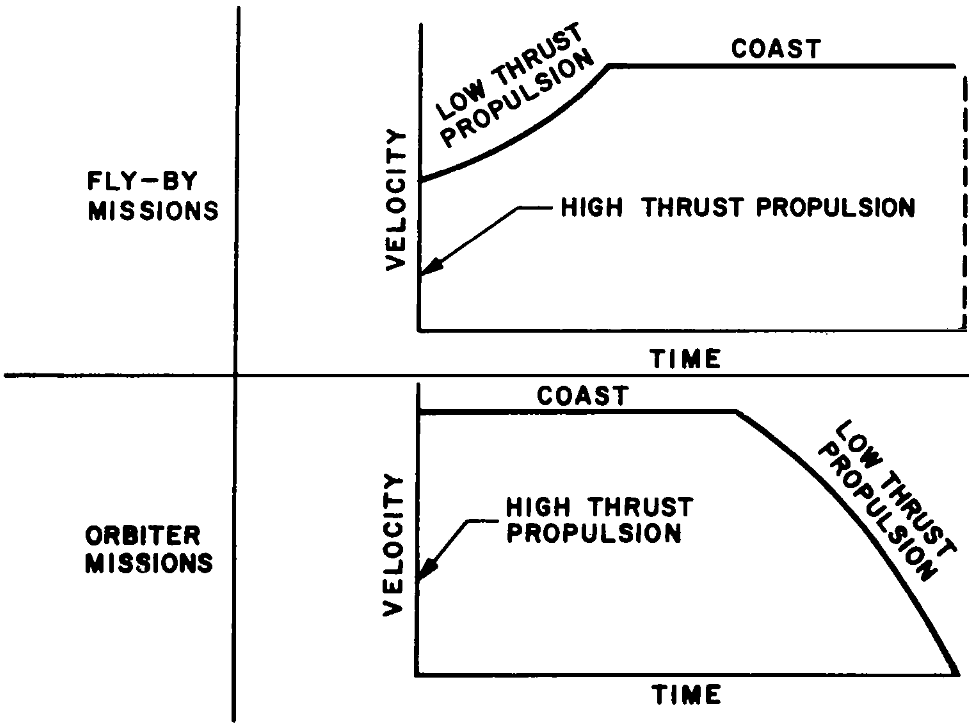

Figure 4-1. One-Dimensional Velocity Diagrams

5.0 PROPULSION SYSTEM CHARACTERISTICS

The NAVIGATOR mission performance results have been based upon current Saturn V estimated payload capabilities and upon projected system characteristics for the additional chemical propulsion stages, the nuclear-electric powerplant, and the electric propulsion systems. The following sections will summarize the assumptions used and the specific system characteristics derived from them.

5.1 CHEMICAL PROPULSION

A two-stage Saturn V booster with a 300 nautical mile orbit payload capability of 240,000 lb has been assumed. One or two transorbital stages using LOX-LH propellants have been assumed to boost the NAVIGATOR vehicle to Earth escape or beyond. Figure 5-1 illustrates the assumed payload capabilities of the resulting three or four stage booster which has been used for the initial gross weight of the nuclear-electric vehicle. These data have been based upon a specific impulse of 450 seconds and a stage mass fraction (λ) of 90%.

The initial hyperbolic excess velocity required to evaluate the low thrust propulsion requirements can be determined from the rocket characteristic velocity by the equation

$$V_o = \sqrt{(V_r^2 + V_e)^2 - 2V_e^2} \qquad (5.1)$$

where V_e is the Earth orbital velocity at the altitude at which the transorbital propulsion takes place. A velocity of 25,000 fps has been used in this study.

5.2 NUCLEAR-ELECTRIC POWERPLANT

The previous NAVIGATOR studies solved for the optimum power level to maximize the payload-trip time relationship for selected levels of powerplant specific weight from 10 to 70 lb/kw. The present study has, however, been restricted to the investigation of the

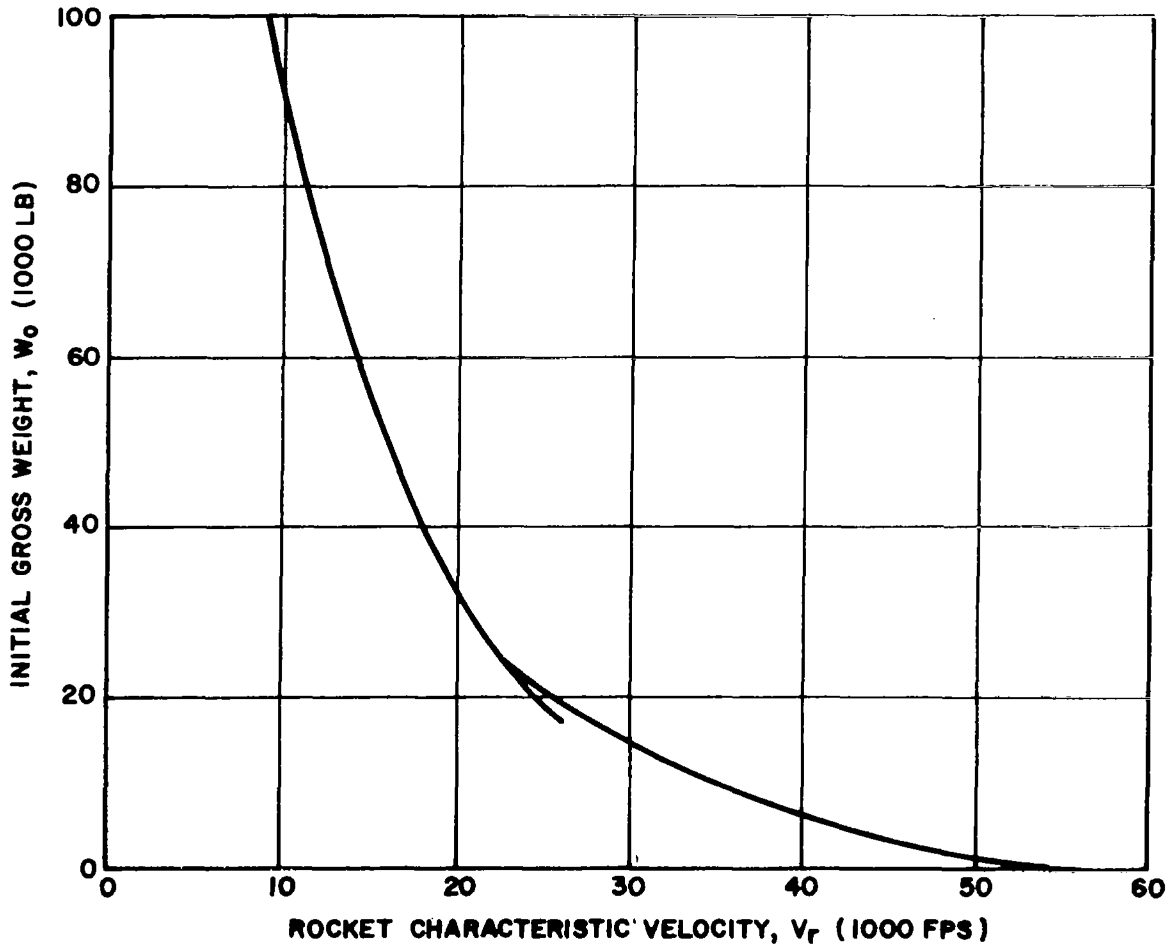

Figure 5-1. Saturn V Booster Capability

performance capabilities at the following power ratings: 100, 160, 240, and 320 kw. This approach has been taken in order to determine the suitability of utilizing a common powerplant for all or most of the NAVIGATOR missions.

Two levels of powerplant specific weight have been considered for each of the above power ratings. These levels correspond to the early and improved technology Potassium-Rankine cycles described in Reference 7. These data are summarized in Figure 5-2. Note that the indicated powerplant specific weight refers to the weight of the powerplant plus shielding per kw of generator power output.

5.3 ELECTRIC PROPULSION SYSTEM

The assumed performance of the electric propulsion system has been based upon projected capabilities of an electron-bombardment ion thrustor using mercury propellant. Figure 5-3 summarizes the specific power characteristics assumed. These data include the thrustor electrical and propellant utilization efficiencies as well as a 96% power conditioning efficiency. The corresponding data used in the previous NAVIGATOR studies is also shown for comparative purposes. Note that the present data indicates improved performance at specific impulse levels above 7,000 seconds and is the result of more recent engine test experience. These data have been represented by the following empirical relationship:

$$
\begin{aligned}
(P/T) &= 40 + .024\, I_{sp} - 3.12\, (10)^{-7}\, I_{sp}^{\ 2}, \text{ kw/lb.} \\
&= A_o + A_1 I_{sp} + A_2 I_{sp}^{\ 2}
\end{aligned}
\qquad (5.2)
$$

The following corresponding relationship has been assumed to represent the specific weight of the thrustor and its associated power conditioning system:

$$w_{th} = 3.0 + 1.18\, (10)^7\, I_{sp}^{-1.808}, \text{ lbs/kw.} \qquad (5.3)$$

Note that the first term represents the power conditioning system weight and the second term the thrustor weight. An additional weight allowance equal to 9% of the propellant weight has been added for the propellant storage and fuel system.

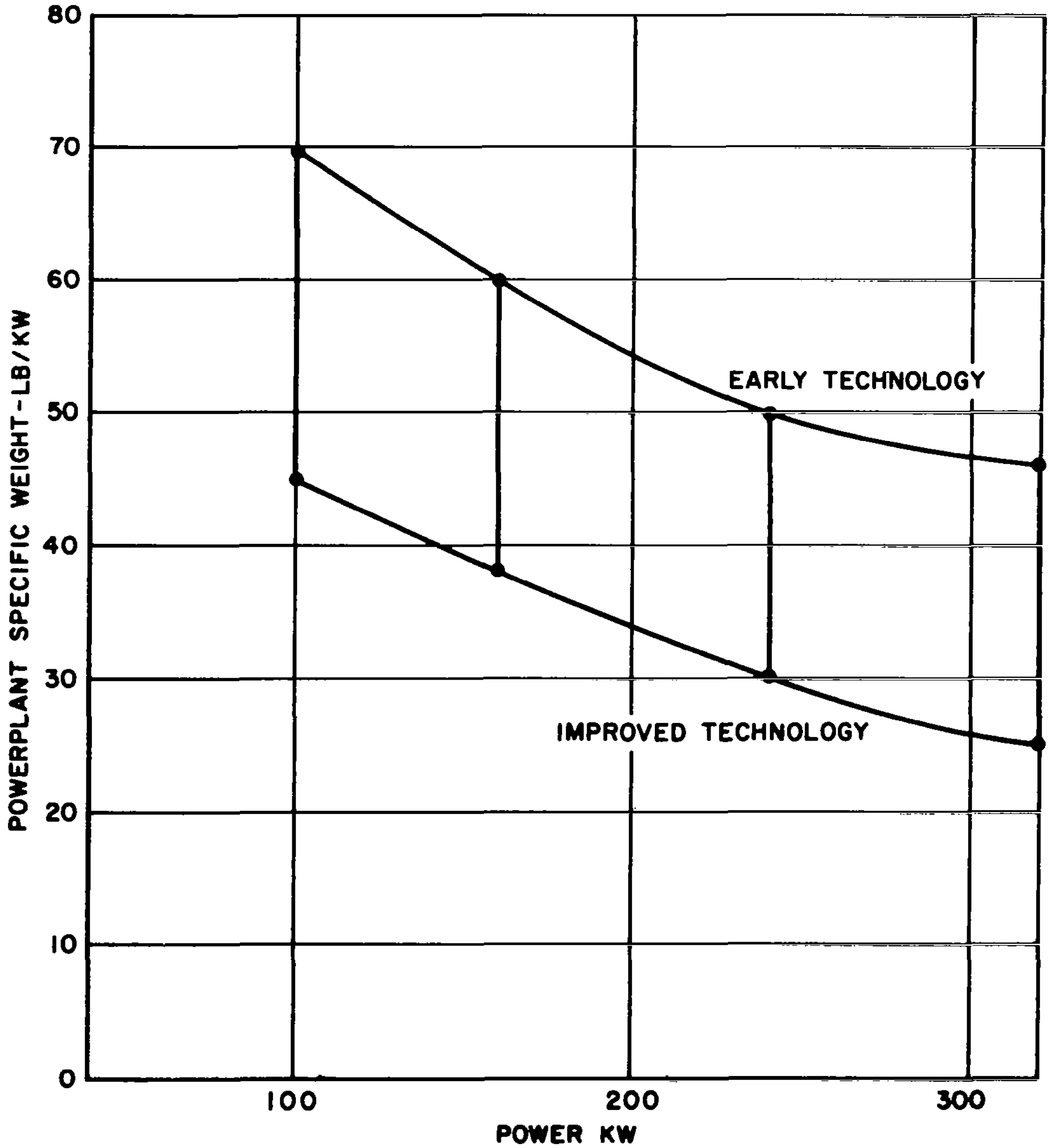

Figure 5-2. Potassium-Rankine Powerplant Specific Weights

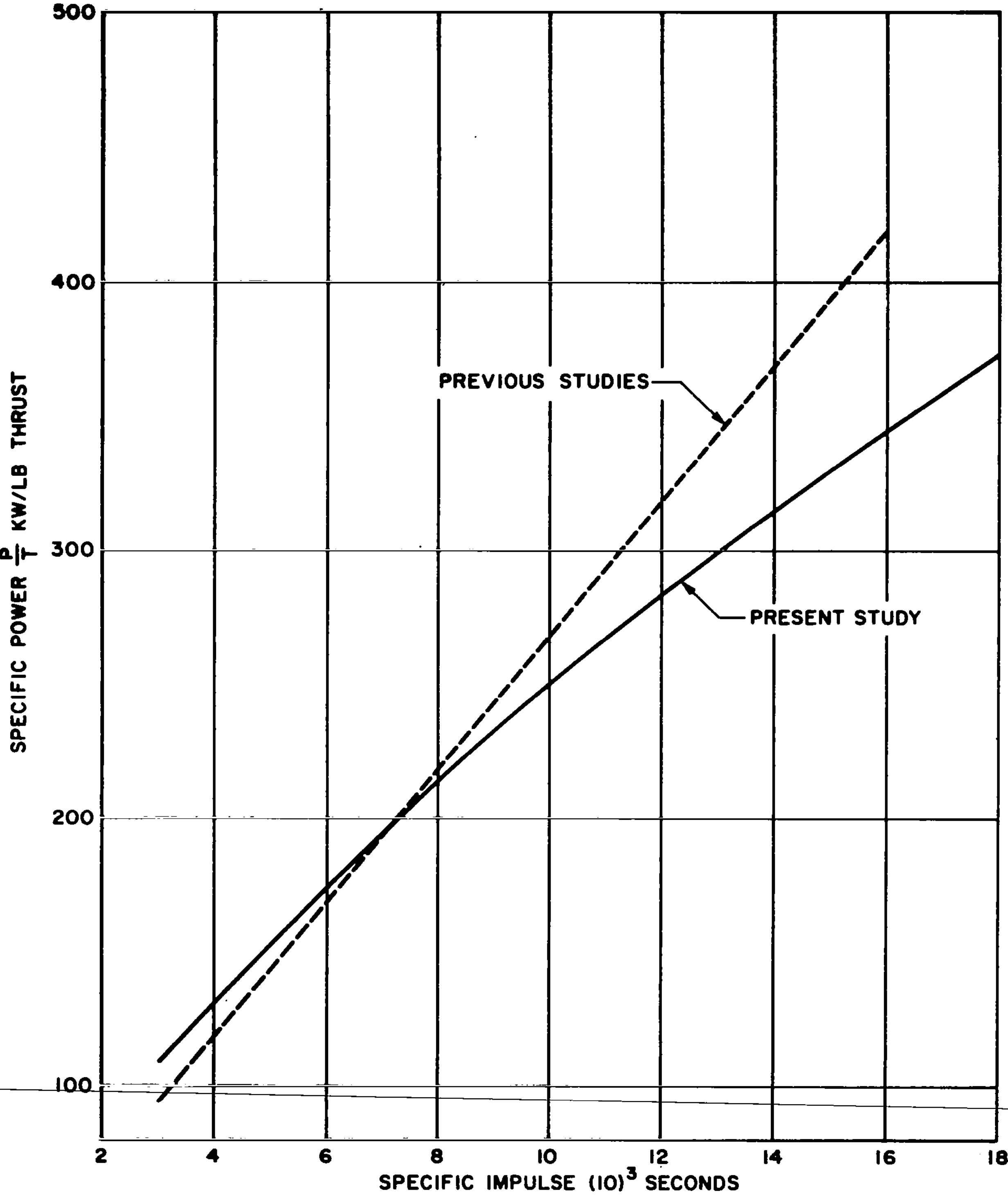

Figure 5-3. Specific Power of Electron Bombardment Thrustor

6.0 MISSION PERFORMANCE ANALYSIS

The results of Sections 4 and 5 have been combined and used to develop a series of mission performance maps for each of the NAVIGATOR fly-by and orbiter missions which illustrate the trade-off between payload capabilities and trip time requirements. Payload has been defined as the terminal spacecraft weight at the completion of electric propulsion minus the weight allocated for propellant tankage and support systems, nuclear-electric powerplant, and electric thrustors and associated power conditioning systems:

$$W_{pl} = \mu W_o - w_t (1-\mu) W_o - wP - w_{th}P \tag{6.1}$$

The payloads presented represent the maximum payload capabilities obtainable for each mission, powerplant, and trip-time combination, and are the result of an optimization process designed to determine the required optimum specific impulse and (high thrust) rocket characteristic velocity.

The following sections describe the specific optimization process used for each of the NAVIGATOR fly-by and orbiter missions. Section 7 contains the resulting mission performance maps.

6.1 FLY-BY MISSIONS

The fly-by mission performance has been based upon a constant propulsion time of 10,000 hours and upon the optimum specific impulse equation of Reference 6:

$$I_{sp} = \sqrt{\frac{-(1 + w_t)\ t_p\ \mu \ln \mu}{(1 - \mu)\ A_1'\ w}} \tag{6.2}$$

This equation can be evaluated for an assumed mass ratio and used to calculate the initial spacecraft weight from the following equations:

$$V_j = gI_{sp} \tag{6.3}$$

$$a_o = (1 - \mu)\frac{V_j}{t_p} \tag{6.4}$$

$$P/T = A_o + A_1 I_{sp} + A_2 I_{sp}^2 \tag{6.5}$$

$$W_o = \frac{gP}{a_o (P/T)} \tag{6.6}$$

The data of Figure 5-1 has been represented by an empirical equation of the form:

$$W_o = B_o + B_1 V_r + B_2 V_r^2 \tag{6.7}$$

Equations (6.6) and (6.7) have been solved simultaneously to obtain the following expression for the rocket characteristic velocity:

$$V_r = \frac{- B_1 - \sqrt{B_1^2 + 4B_2(W_o - B_o)}}{2B_2} \tag{6.8}$$

The resulting characteristic velocity can be used to obtain the initial hyperbolic excess velocity V_o from equation (5.1).

The total trip time can then be obtained from a simultaneous solution of equations (4.7) and (4.10):

$$t_t = \frac{B - \sqrt{B^2 - 4AC}}{2A} \tag{6.9}$$

where:

$$A = \frac{L_o - L_m}{t_m^2} \tag{6.10}$$

$$B = V_o + \frac{2(L_o - L_m)}{t_m} - V_j \ln \mu \tag{6.11}$$

$$C = L_o - V_j t_p - \frac{V_j^2}{a_o} \ln \mu \tag{6.12}$$

The resulting equations (6.1) through (6.12) constitute the calculation procedure used in obtaining the final fly-by performance maps where a range of mass ratio values have been used to generate the desired variation in trip time.

6.2 MAJOR PLANET ORBITERS

The orbiter missions differ from the fly-by missions in the requirement for a parabolic approach to the target planet at the completion of the heliocentric propulsion period and the requirement for subsequent planetary propulsion to achieve the desired terminal planetary orbit. The requirement for parabolic approach in conjunction with the use of a single electrical propulsion period imposes the following constraining relationship upon the initial velocity:

$$V_o = -V_j \ln \mu_h \tag{6.13}$$

where the subscript h refers to the heliocentric phase of the mission. The orbiter propulsion times have been allowed to vary in order to maintain the freedom to optimize the specific impulse and the rocket characteristic velocity.

Two approximations were introduced into the analysis in order to achieve a capability for a direct graphical optimization process that would not require the use of equation (6.2). The first of these has been the use of the series expansion for the heliocentric mass ratio:

$$\mu_h = e^{-(V_o/V_j)} = 1 - \frac{V_o}{V_j} + \frac{V_o^2}{2V_j^2} \qquad (6.14)$$

It is estimated that the maximum error introduced by the use of only three terms is of the order of 1.5%. The second approximation involved replacement of the quadratic specific power equation (5.2) by a linear equation. This equation has been combined with equation (6.3) to obtain

$$\frac{P}{a_o W_o} = 6.9(10)^{-4} + 1.094(10)^{-8} V_j \qquad (6.15)$$

$$= A_o' + A_1' V_j$$

Equations (4.11), (6.13), (6.14), and (6.15) were then combined to obtain the following quadratic equation in jet velocity:

$$V_j^2 + \left[\frac{2P}{A_1' W_o V_o} \left(\frac{L}{V_o} - t_h \right) + \frac{A_o'}{A_1'} - V_o \right] V_j - \frac{A_o'}{A_1'} V_o = 0 \qquad (6.16)$$

Note that the approximations associated with equations (6.14) and (6.15) have been used only in the above jet velocity calculation, and that the complete equation (5.2) has been used in all subsequent calculations.

Equations (4.7), (6.16), and (6.14) can then be used to calculate the characteristic length, jet velocity, and heliocentric mass ratio as a function of an assumed heliocentric trip time and rocket characteristic velocity. These results must then be used to calculate the terminal planetary propulsion requirements from the equations of Reference 6:

$$\mu_{pl} = e^{-V_{pl}/V_j} \tag{6.17}$$

$$t_{pl} = (1 - \mu_{pl}) \frac{\mu_h V_j}{a_o} \tag{6.18}$$

The total mission requirements can then be obtained from the following:

$$t_t = t_h + t_{pl} \tag{6.19}$$

$$\mu = \mu_h \mu_{pl} \tag{6.20}$$

Equation (6.1) can then be used to determine the final payload.

The resulting procedure, therefore, constitutes the major planet orbiter calculation procedure. Figure 6-1 illustrates the result of a series of calculations for a Jupiter II orbiter mission with a 100 kw powerplant. Payload has been plotted against trip time for lines of constant heliocentric trip time. Note that characteristic velocity has been varied along each trip time line. Optimum performance is represented by the envelope of maximum payload at constant trip time. Envelopes of this type have been defined for each of the major planet orbiter missions and were used to generate the final performance maps of Section 7.

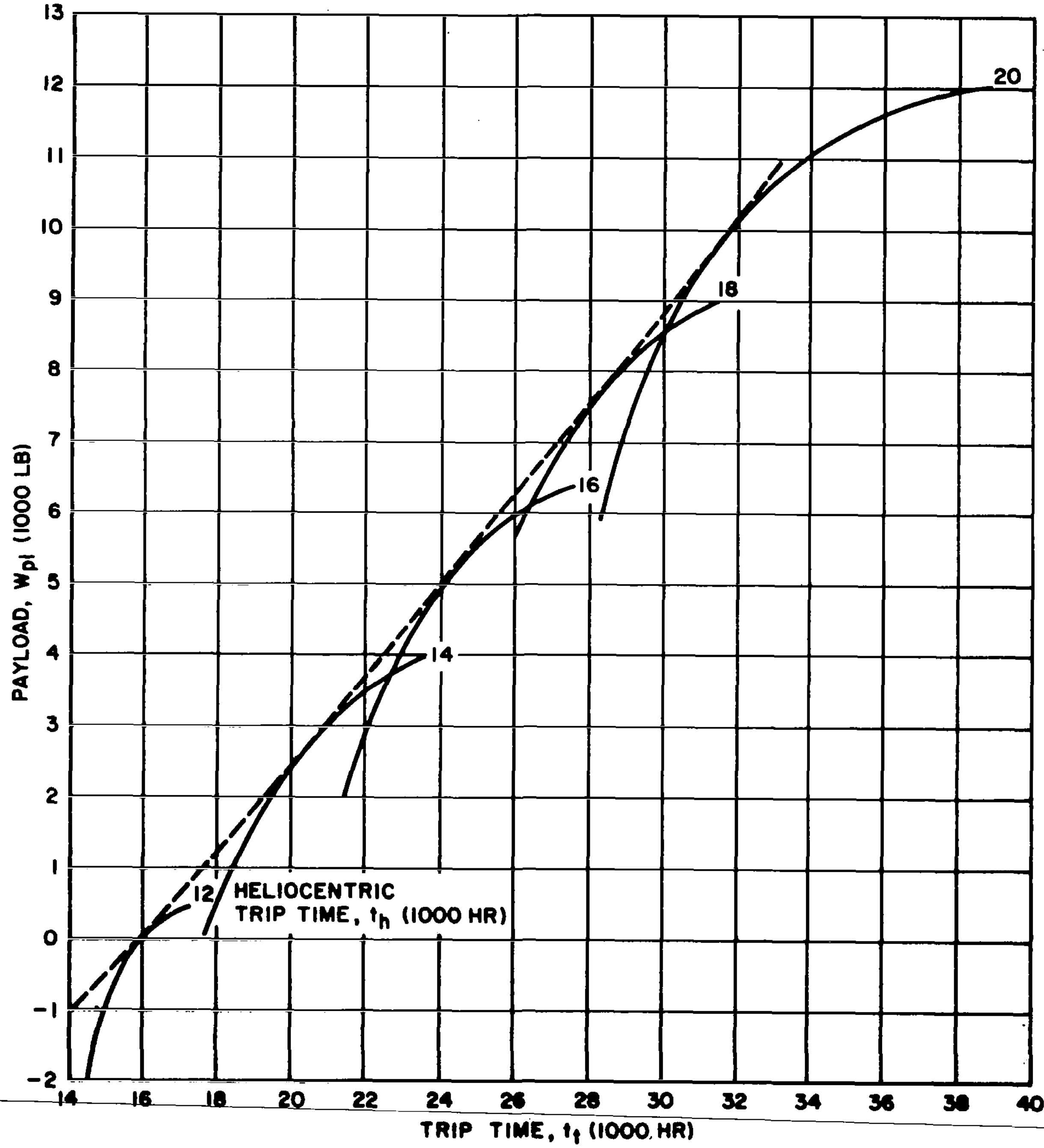

Figure 6-1. Jupiter II Orbiter Optimization

6.3 MINOR PLANET ORBITERS

The use of the preceding approach for the minor planet orbiters indicated that the optimum payload-trip time envelope occurred at specific impulse levels well below the 3,000 second level believed to be the lower limit for current state-of-the-art thrustors. The minor planet procedure was, therefore, modified to eliminate the graphical optimization process and to calculate performance for operation only at 3,000 seconds specific impulse. This simplification would appear to permit a direct calculation for the final minor planet orbiter performance. This situation is complicated, however, by the relatively small range of characteristic velocities in which there is sufficient propulsion time available to achieve the required parabolic approach to the target planet. This limitation has been evaluated by the following analysis.

The variation of heliocentric propulsion time with characteristic velocity for a specified power level and specific impulse can be calculated from equations (5.1), (6.7), (6.14), and the following additional equations:

$$a_o = \frac{Pg}{W_o(P/T)} \tag{6.21}$$

$$t_{ph} = (1 - \mu_h) \frac{V_j}{a_o} \tag{6.22}$$

The minimum characteristic length that can be generated for a given characteristic velocity can then be obtained by setting the heliocentric trip time equal to the heliocentric propulsion time thereby reducing equation (4.11) to the following:

$$L = V_j \left(\frac{V_j}{a_o} - t_{ph} \right) \tag{6.23}$$

Figure 6.2 summarizes the results of calculations based upon the above relationships for power levels of 100, 160, 240, and 320 kw. Superimposed on these data are the characteristic length requirements for the Mercury, Venus, and Mars orbiter missions as obtained from equation 4.7. Points of intersection between constant power and mission requirement lines indicate points where operation would involve continuous propulsion (no coast). Feasible operation is not possible at those characteristic velocities where the constant power line is above the mission requirement line. Feasible operation is possible at all other characteristic velocities with a finite coast period. Venus missions, for example, would be limited to operation at characteristic velocities below 16,000 fps at 320 kw and below 13,000 fps at 100 kw.

The above results are, however, based upon the use of a fully loaded Saturn V booster and upon the use of an ion engine limited to operation at or above 3,000 seconds specific impulse. The above conclusions can, therefore, be changed by off-loading the Saturn V booster, by switching to a Saturn IB booster, or by using thermal arc jet propulsion which would permit operation down to specific impulse levels of the order of 800 to 1,000 seconds. These options were, however, considered to be beyond the scope of the current study. The final performance maps were, therefore, developed for the feasible operating region indicated by Figure 6-2.

The minor planet performance calculations utilize equations (6.21) and (6.22) to calculate the initial acceleration and the heliocentric propulsion time. The heliocentric trip time is then obtained from a simultaneous solution of equations (4.7) and (4.11):

$$\left[\frac{L_o - L_m}{t_m^{\,2}}\right] t_h^{\,2} - \left[V_o + \frac{2(L_o - L_m)}{t_m}\right] t_h + L_o + (V_o + V_j)t_{ph} - \frac{V_o V_j}{a_o} = 0 \qquad (6.24)$$

The remainder of the procedure involving the calculation for the terminal planetary mass ratio and propulsion time is identical to the procedure described in the preceding section on the major planet orbiters.

The results of the minor planet orbiter performance calculations are summarized in Section 7.

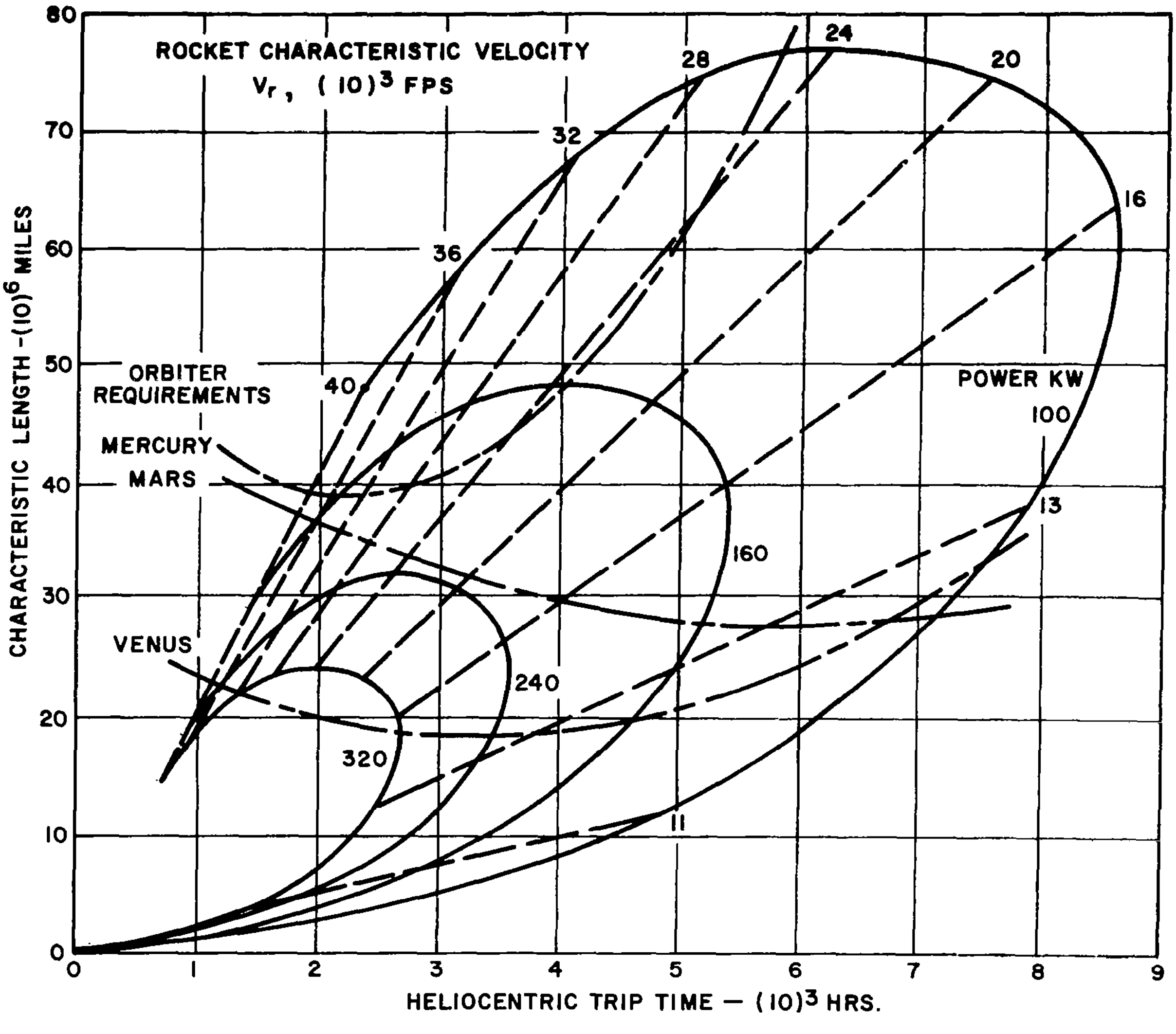

Figure 6-2. Minimum Characteristic Length for Operation at 3,000 Seconds Specific Impulse

7.0 MISSION PERFORMANCE

This section summarizes the performance and propulsion requirements for the various NAVIGATOR missions considered. They were determined as a result of the optimization processes outlined in the preceding section.

In the previous NAVIGATOR study (see Reference 8) an analysis was made to determine the minimum useful scientific payload required for each of the missions. The results of the present study indicate that most of these payloads can be delivered in less than 100,000 hours.

7.1 FLY-BY MISSIONS

The fly-by performance is shown in Figures 7-1 through 7-3. For these missions, the propulsion time was assumed to be 10,000 hours.

Figure 7-1 shows that all of the missions investigated can be performed with an early technology powerplant at all four power levels in less than 50,000 hours total trip time. If the payload is substantially increased above the minimum acceptable value, the lower power Neptune and Pluto missions appear to be unfeasible due to assumptions made in the performance optimization.

Figure 7-2 indicates that trip times may be reduced from 10 to 25 percent with improved technology. Note that the Saturn mission at higher power levels does not require the full 10,000 hours of propulsion time.

It is seen in both Figures 7-1 and 7-2 that the optimum power is above 320 kw for all of the fly-bys.

Figure 7-3 indicates that improving the level of technology results in a lower nuclear-electric powerplant weight thereby allowing higher values of the chemical rocket characteristic velocity. In addition, the improved technology allows higher specific impulse values.

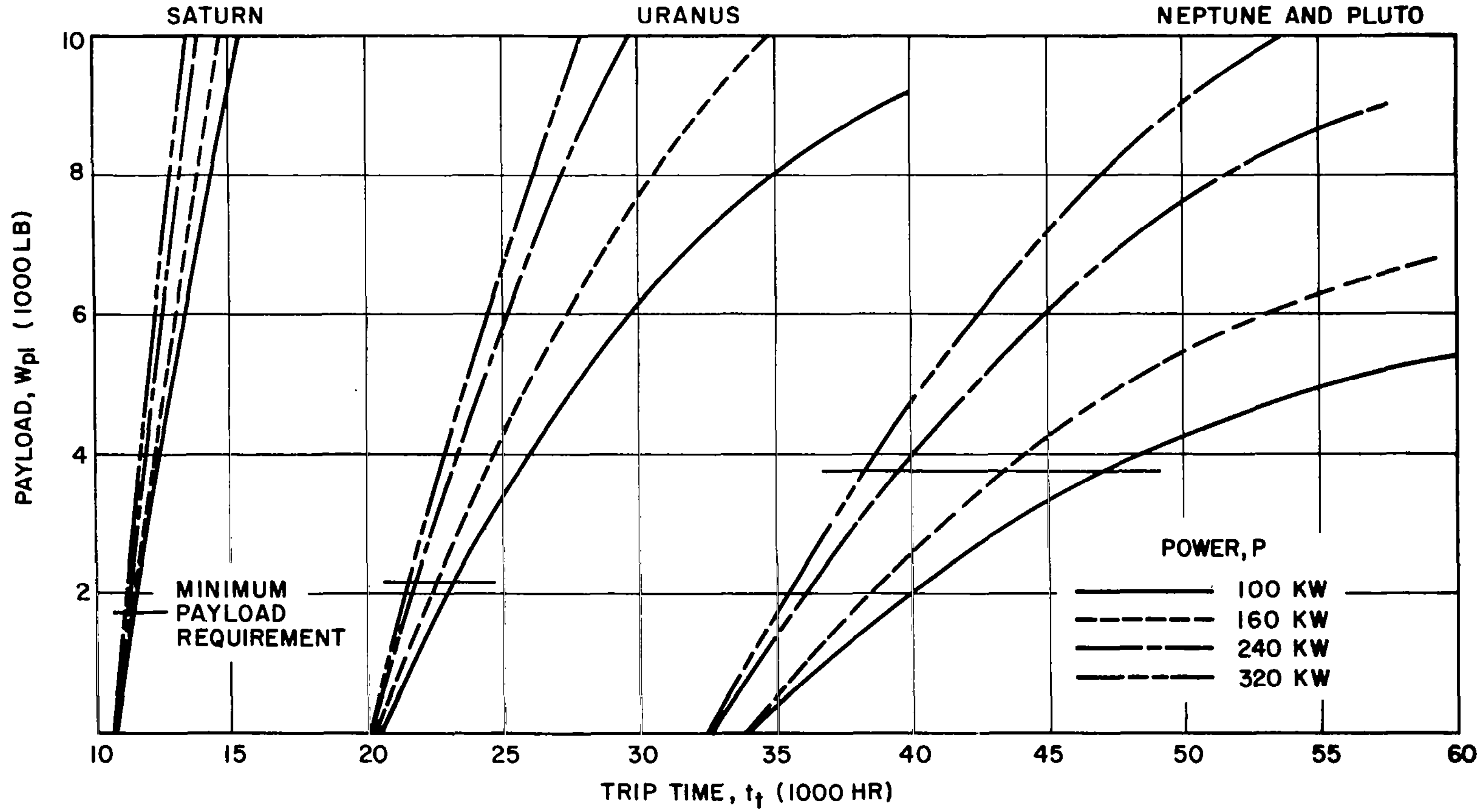

Figure 7-1. Major Planet Fly-by Payload Capability with Early Technology

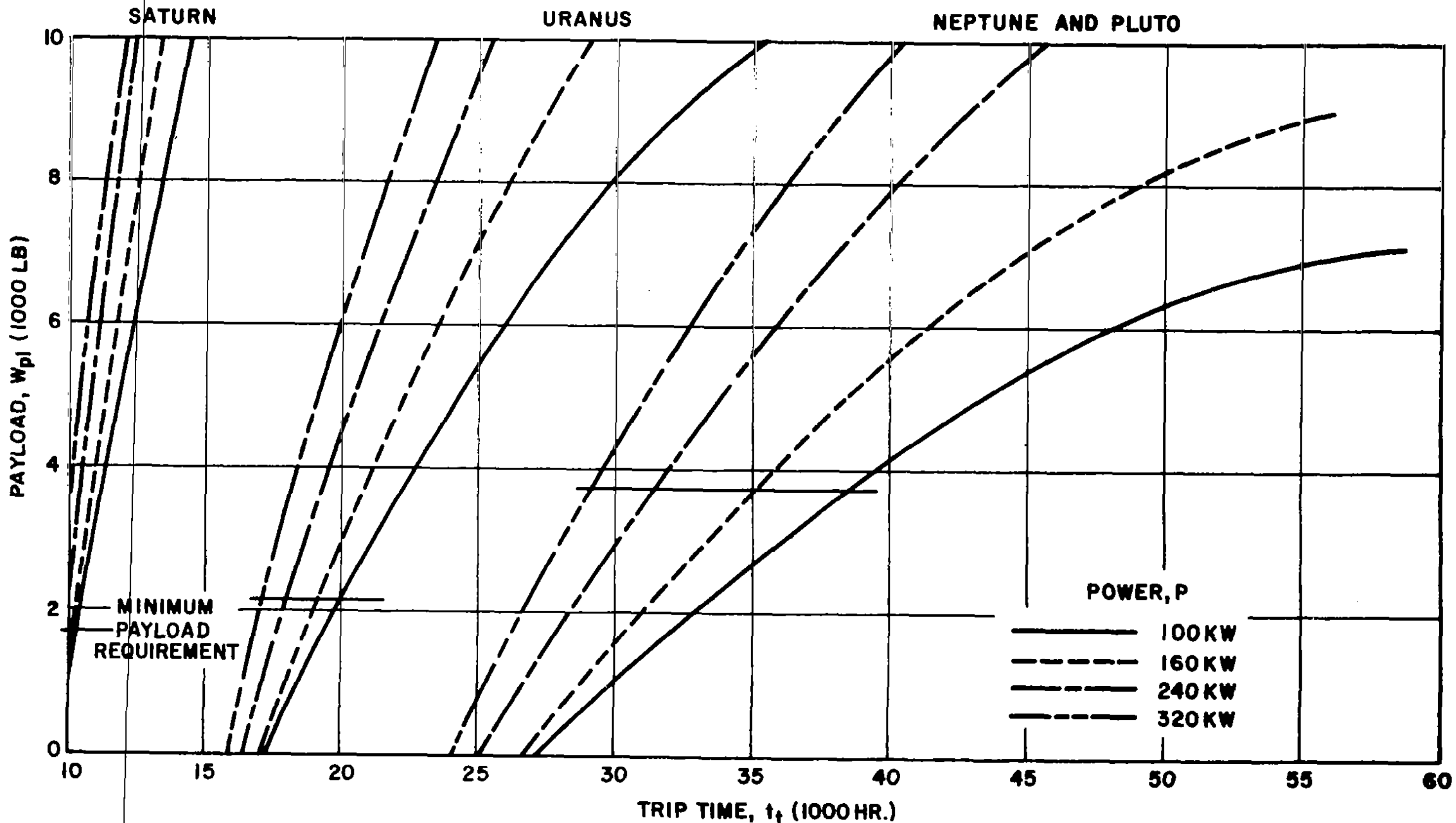

Figure 7-2. Major Planet Fly-by Payload Capability with Improved Technology

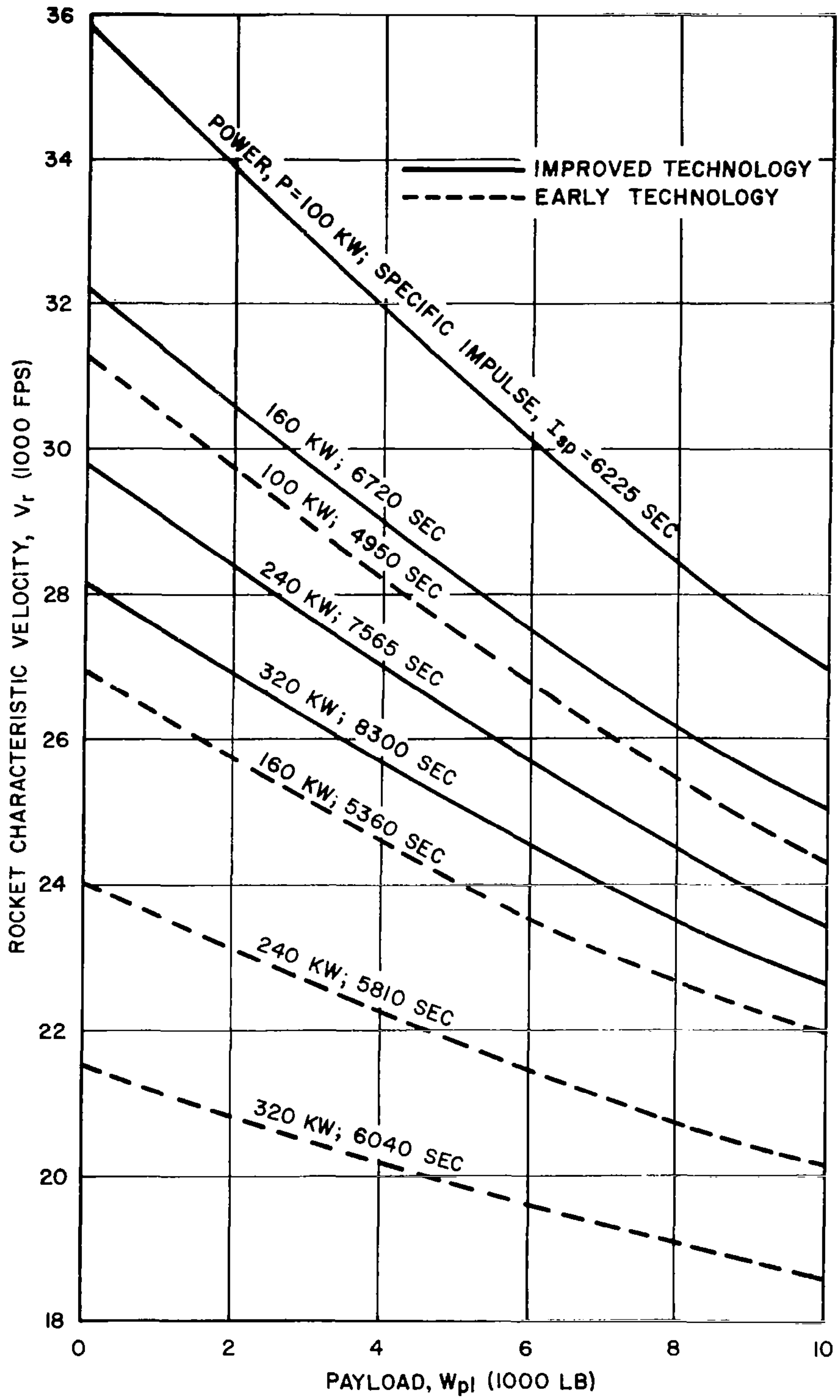

Figure 7-3. Rocket Characteristic Velocity Variation and Specific Impulse Levels for Fly-bys

For a given power and powerplant, specific impulse was found to be essentially constant, not varying more than 500 seconds. Note that for a given specific weight-power combination, the rocket characterisitc velocity required to deliver a specified payload is independent of the mission requirements.

7.2 MAJOR PLANET ORBITERS

Figures 7-4 through 7-9 summarize performance and powerplant requirements for the major planet orbiters.

The payloads of Figures 7-4 and 7-5 are presented only up to 100,000 hour trip time. With early technology it was not possible to attain the minimum acceptable payload requirements for any of the Uranus, Neptune, or Pluto missions or the Saturn II mission at 320 kw within this trip time bound. By improving the technology level, all Saturn II missions and the lower power Uranus missions could be performed in 100,000 hours. Note the trend of the optimum power level to shift downward as the difficulty of the mission increases.

Figure 7-6 shows the propulsion time requirement; it is independent of technology level a given trip time. The improved technology, however, reduces the trip time and hence reduces the propulsion time as indicated in Figure 7-6. The reductions in trip time and propulsion time ranged from 12 to 35 percent. The percentage reduction increased with increasing power.

The variation of specific impulse with power is given in Figure 7-7. This parameter did not vary significantly with trip time. Specific impulse was found to increase with increasing power and with the difficulty of the mission. Going from early to improved technology resulted in a decrease; the change was negligible for Jupiter I.

The optimum rocket characteristic velocity requirements for each of the missions is shown in Figures 7-8 and 7-9. Note the consistent decrease of this requirement with increasing trip time and power level.

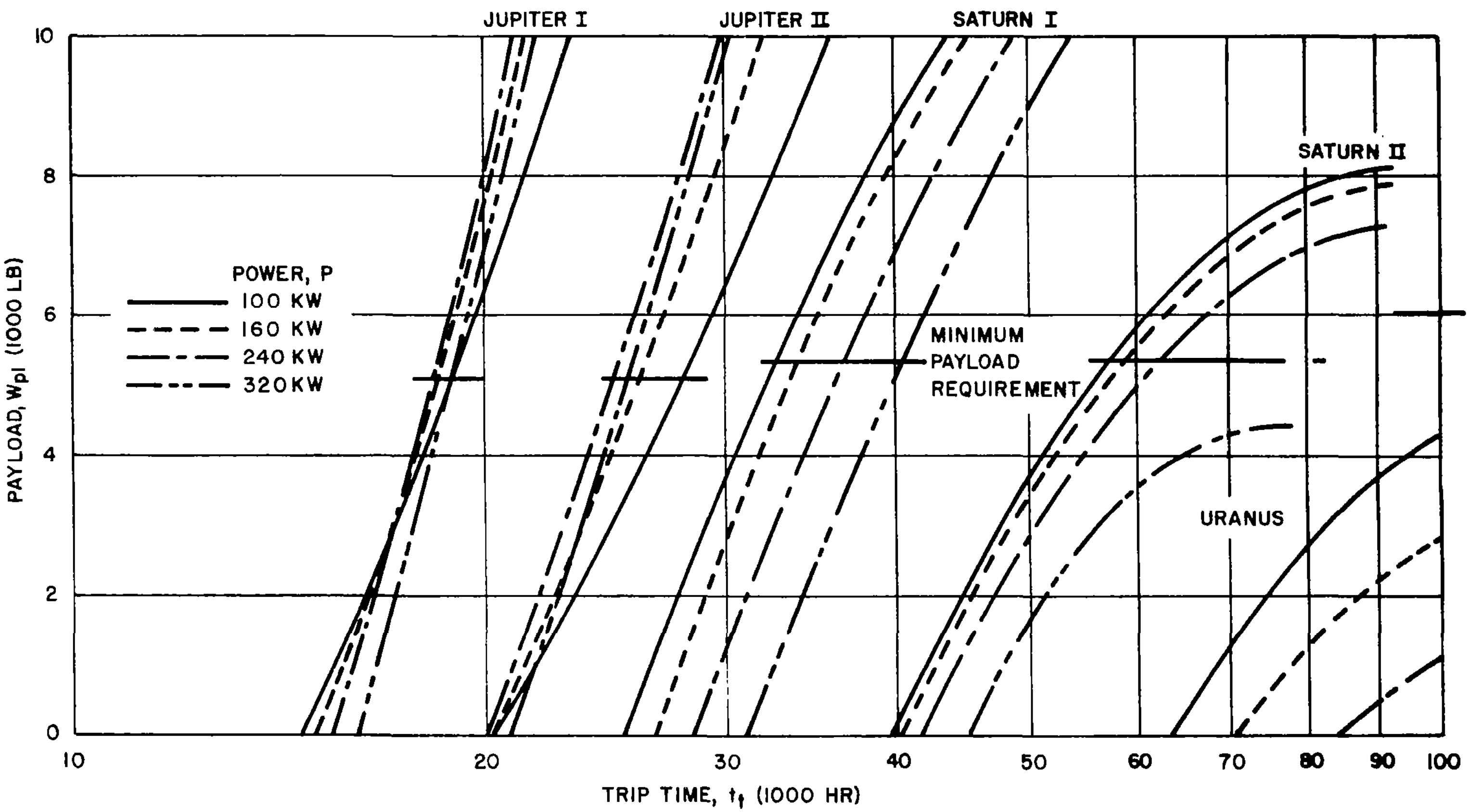

Figure 7-4. Major Planet Orbiter Payload Capability with Early Technology

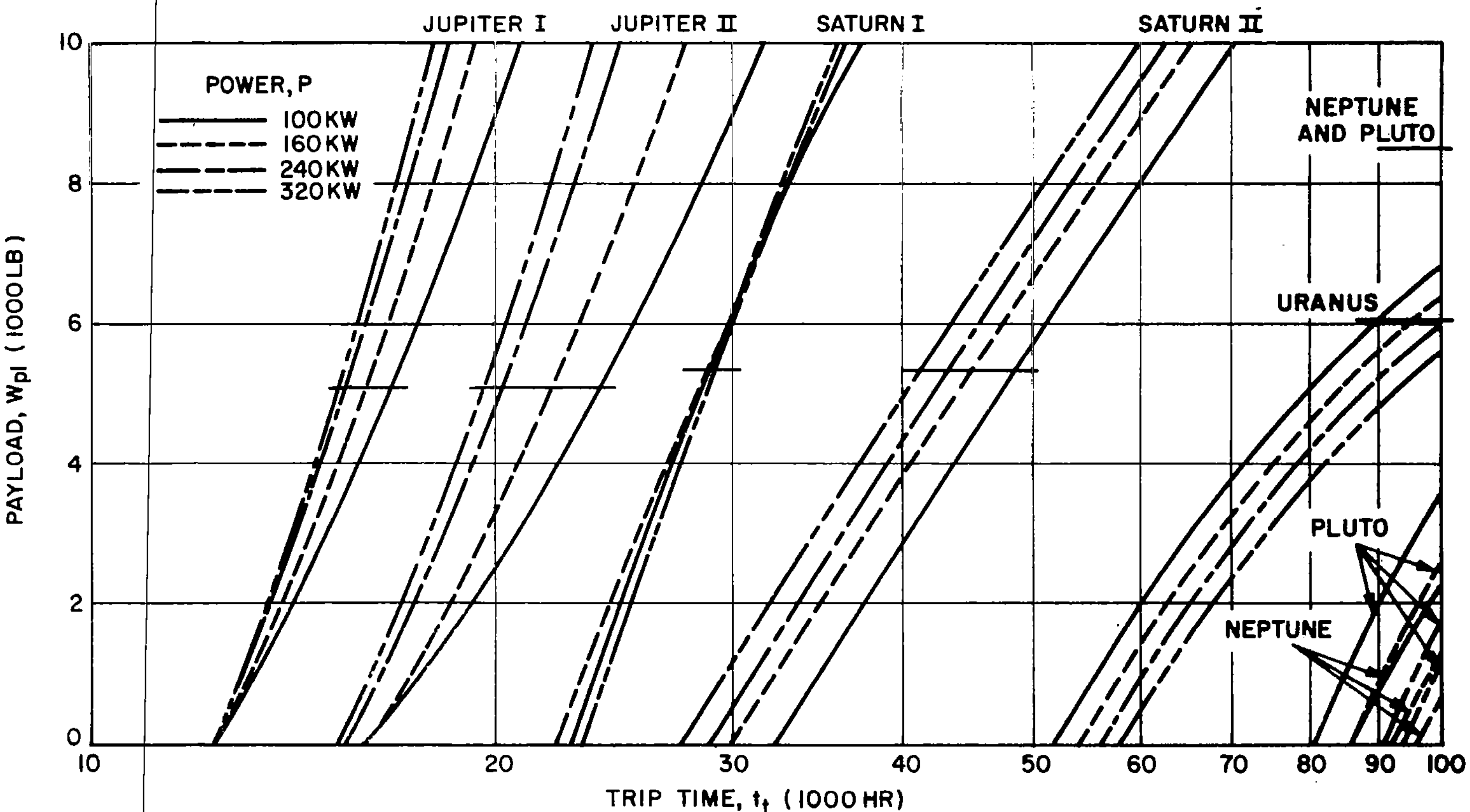

Figure 7-5. Major Planet Orbiter Payload Capability with Improved Technology

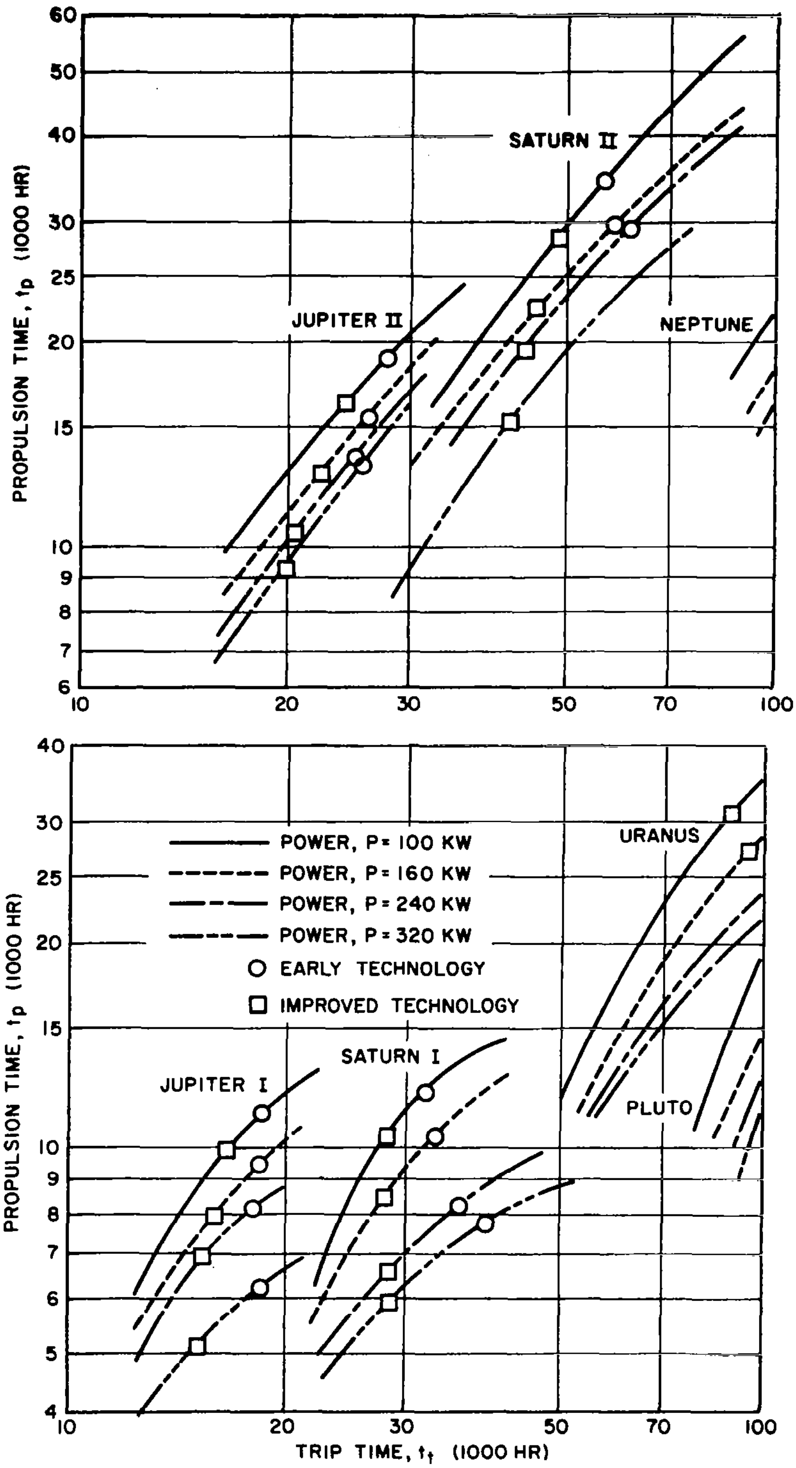

Figure 7-6. Major Planet Orbiter Propulsion Time Requirements

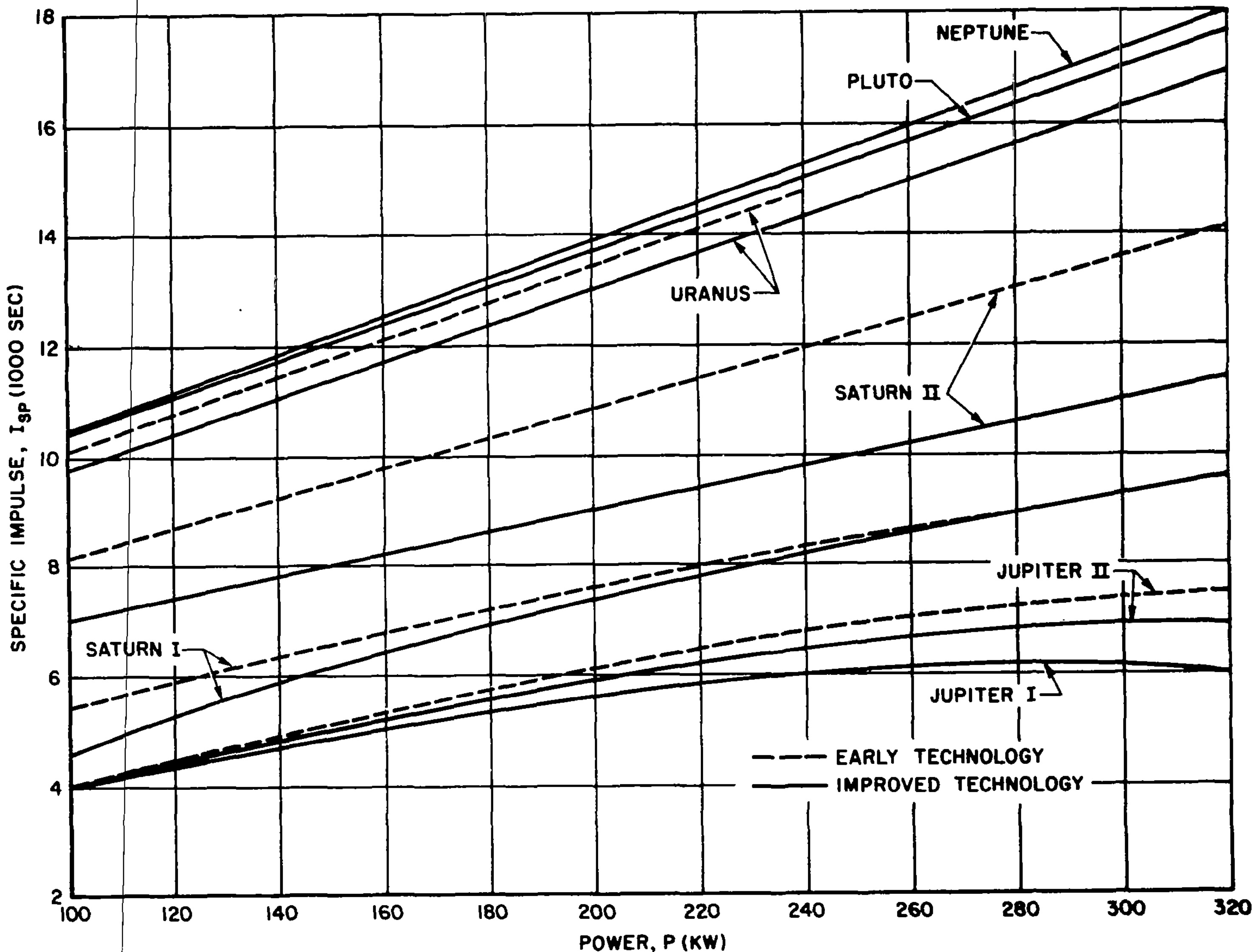

Figure 7-7 Major Planet Orbiter Specific Impulse Requirements

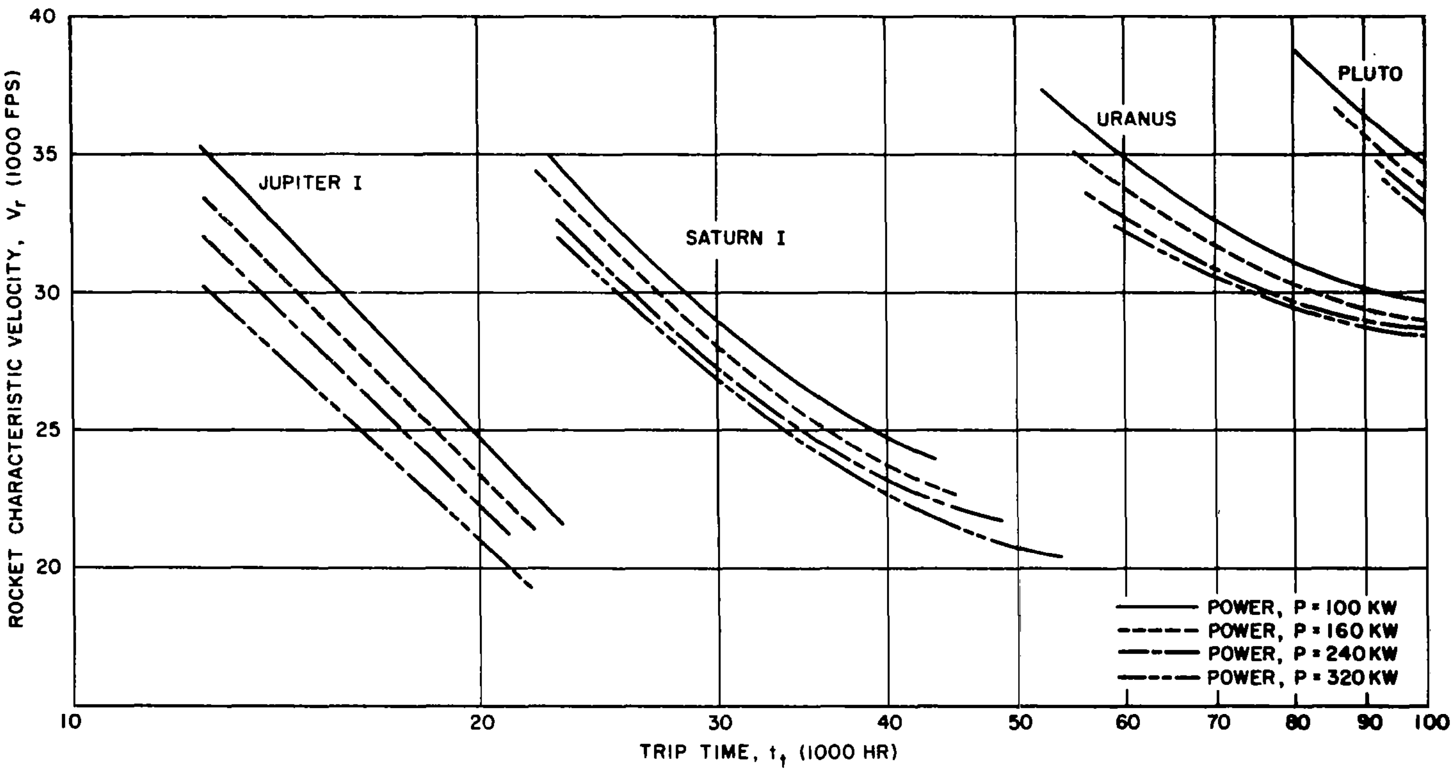

Figure 7-8. Rocket Characteristic Velocity for Major Planet Orbiters

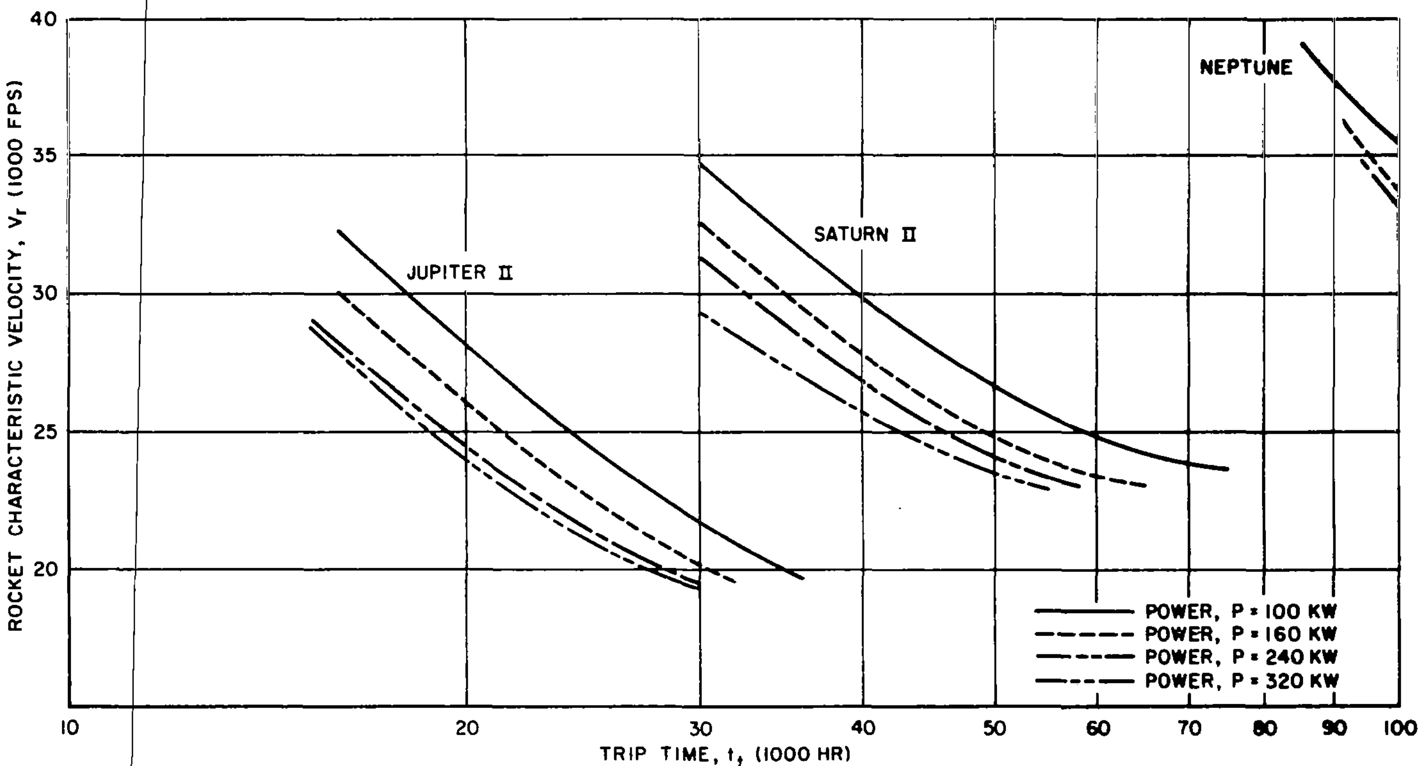

Figure 7-9. Rocket Characteristic Velocity for Major Planet Orbiters (Continued)

7.3 MINOR PLANET ORBITERS

For the Saturn V booster characteristics assumed in this study, application of the optimization process used for the major planet orbiters resulted in specific impulse levels well below those of the assumed nuclear-electric propulsion system. Hence, the optimization process was modified and the specific impulse constrained to remain at 3,000 seconds. The performance results obtained from this procedure are presented in Figure 7-10 through 7-14.

Figures 7-10 through 7-12 show the relationship between payload, trip time, and power for each of the three planets. The simulated "three dimensional" maps shown were intended to illustrate this relationship more effectively. The lower and upper surfaces in each figure represent levels of early and improved technology respectively; these are intersected by planes of constant power. It is seen that for Venus and Mars the payloads are generally higher than those of interest. For Venus it is clear that with a Saturn V booster, higher power levels would be required to reduce the payload to a reasonable value; for both missions it is probable that a smaller booster would be desirable. The payloads for Mercury, on the other hand, are within the range of interest, although the optimum power is greater than 320 kw.

The propulsion time requirements are illustrated in Figure 7-13 and the rocket characteristic velocity in Figure 7-14. It is seen in Figure 7-13 that some of the missions hit a continuous propulsion constraint. This is due to the fact that for these missions the range of rocket characteristic velocities allowing sufficient propulsion time for parabolic planetary approach is relatively small as indicated in Figure 7-14. As in the case of the major planet orbiters, the rocket characteristic velocity is decreasing with increasing trip time, and the propulsion time is decreasing with increasing power.

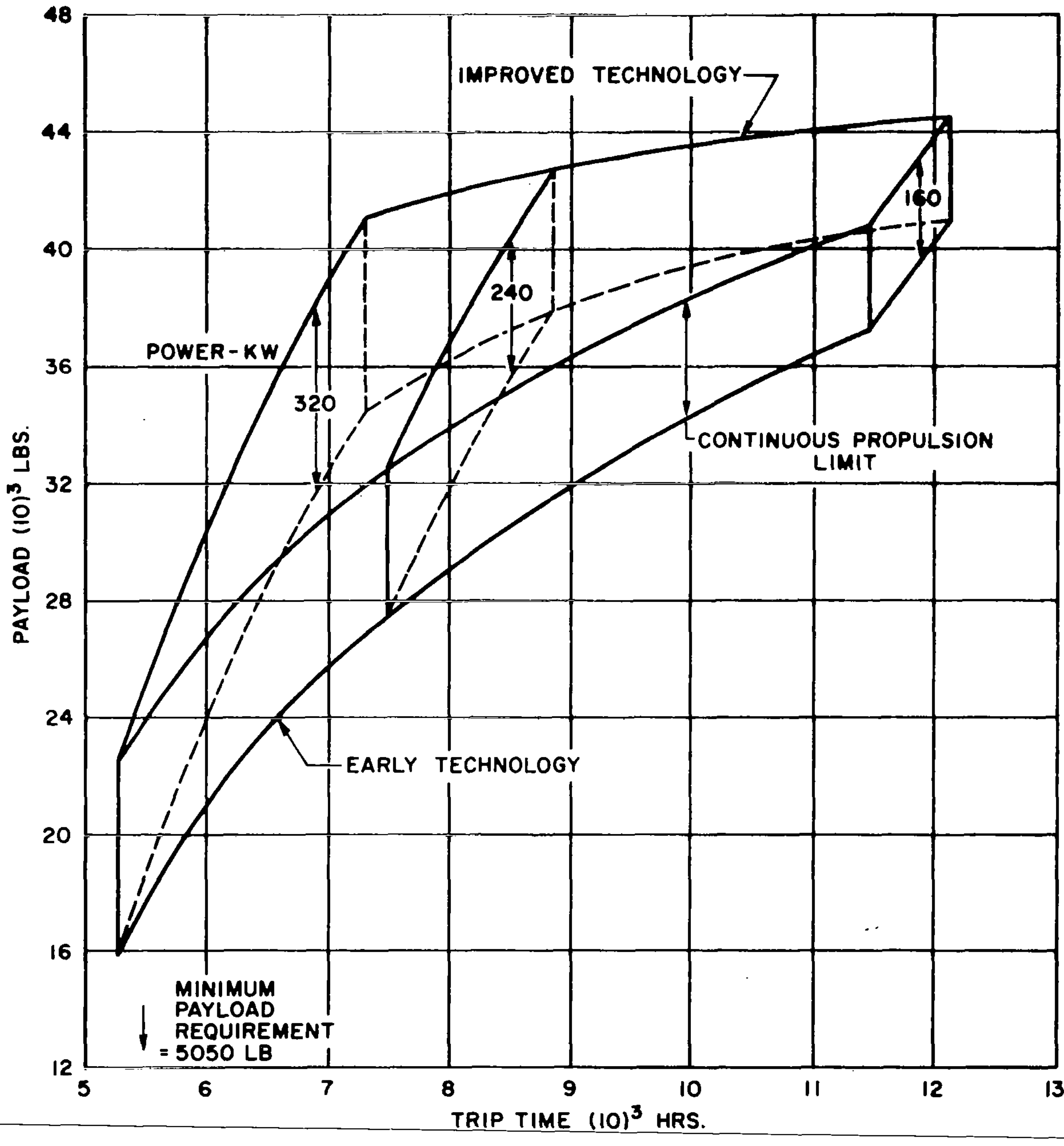

Figure 7-10. Venus Orbiter Mission Performance

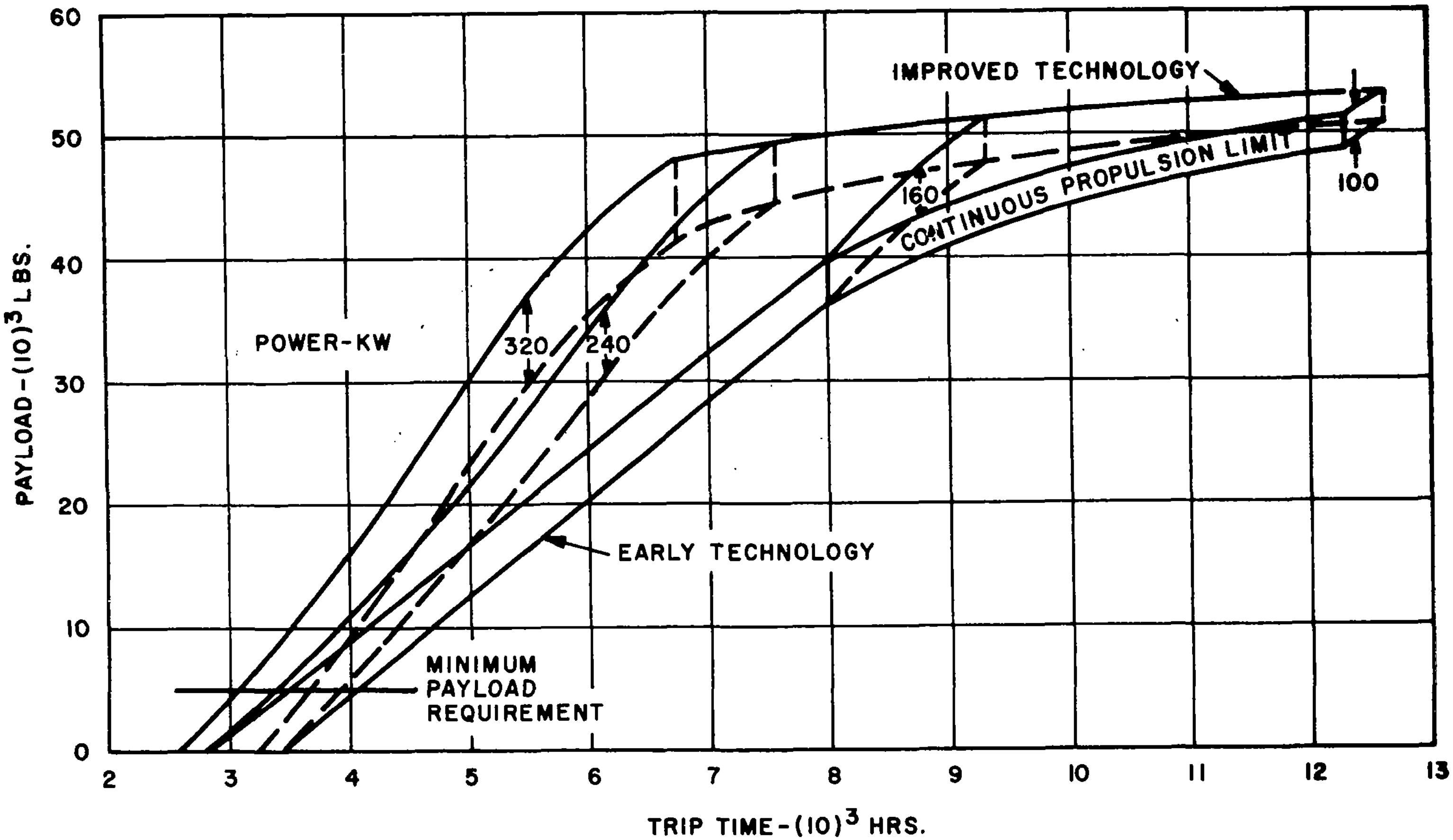

Figure 7-11. Mars Orbiter Mission Performance

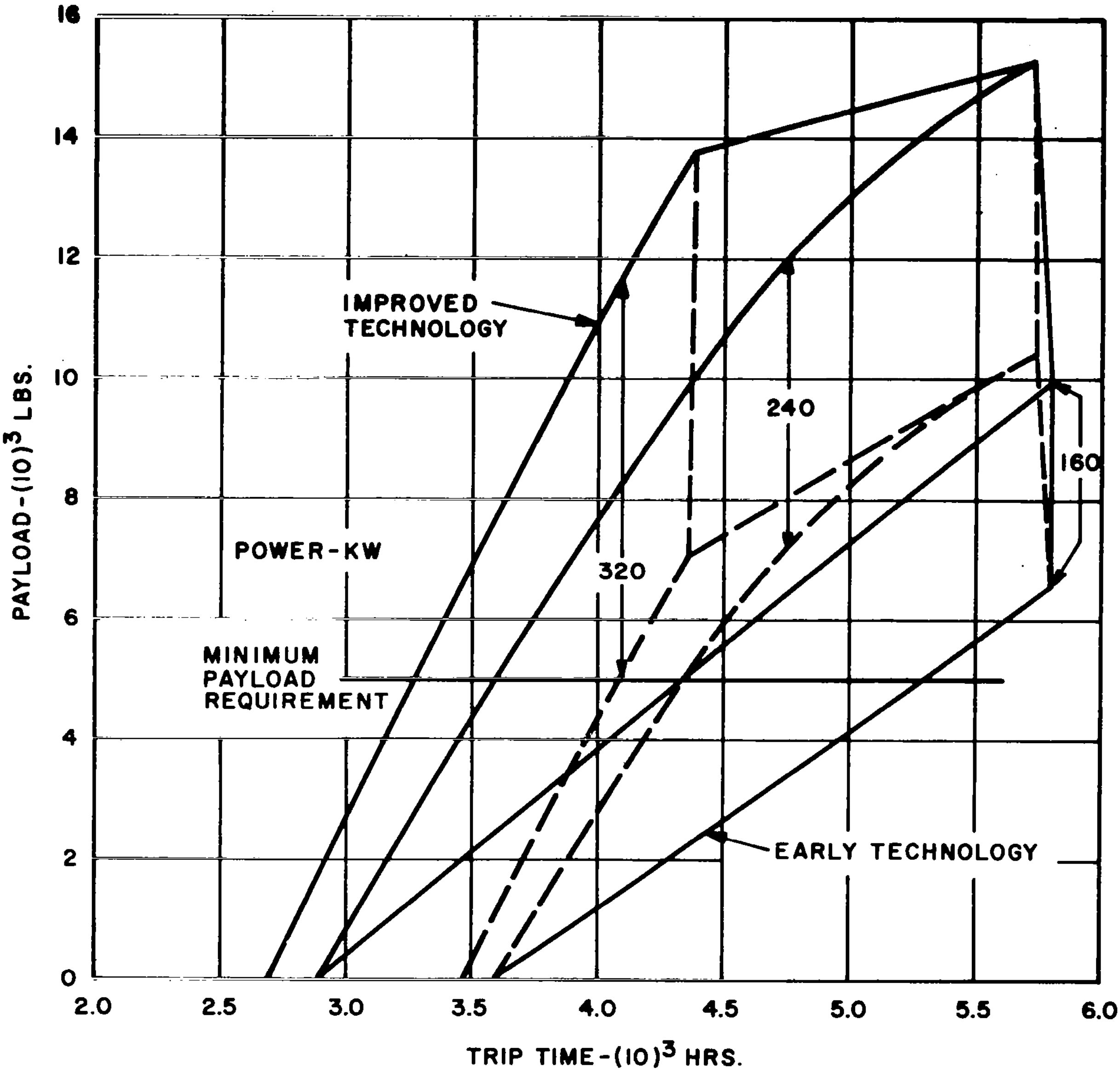

Figure 7-12. Mercury Orbiter Mission Performance

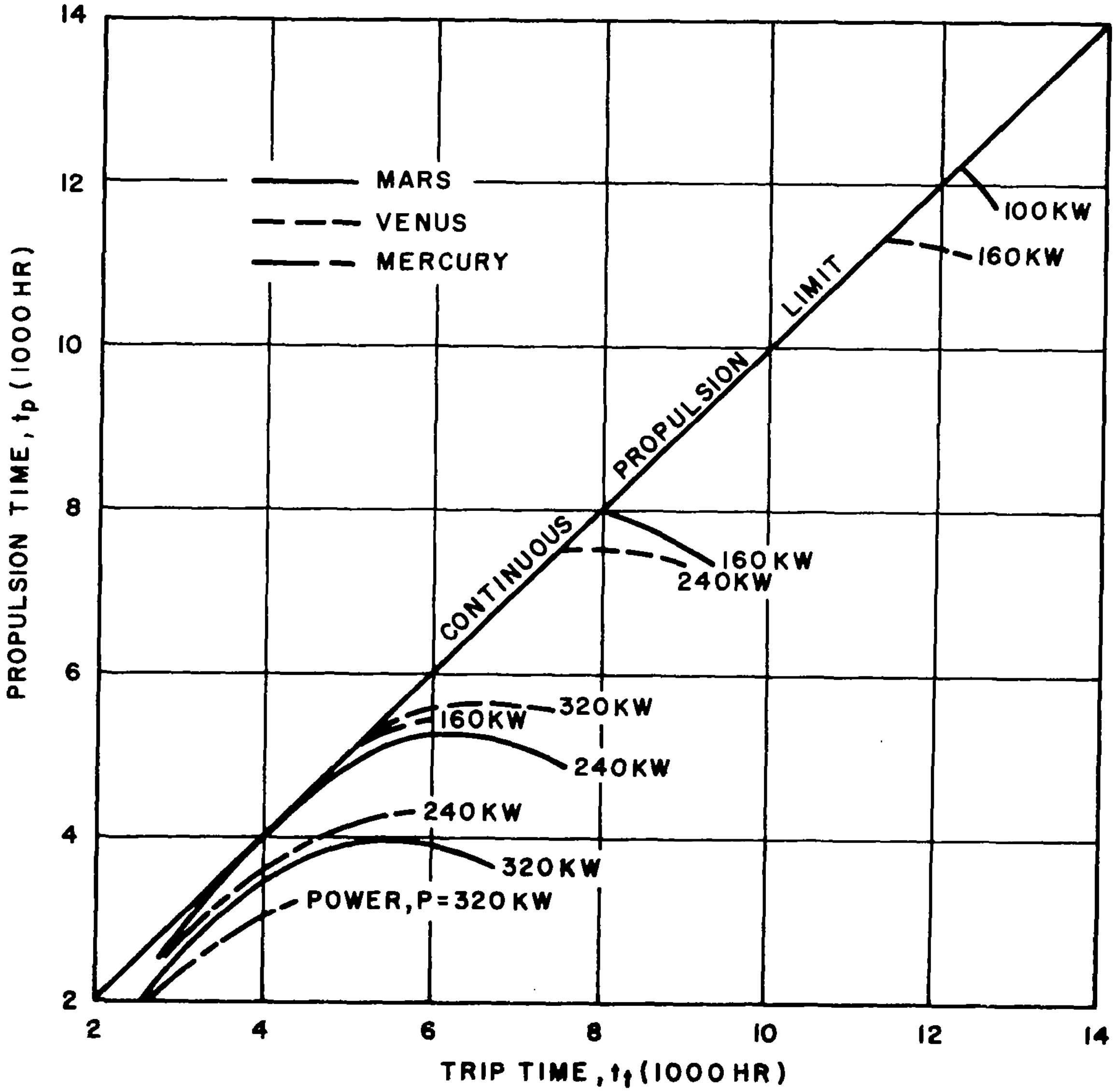

Figure 7-13. Minor Planet Orbiter Propulsion Time Requirements

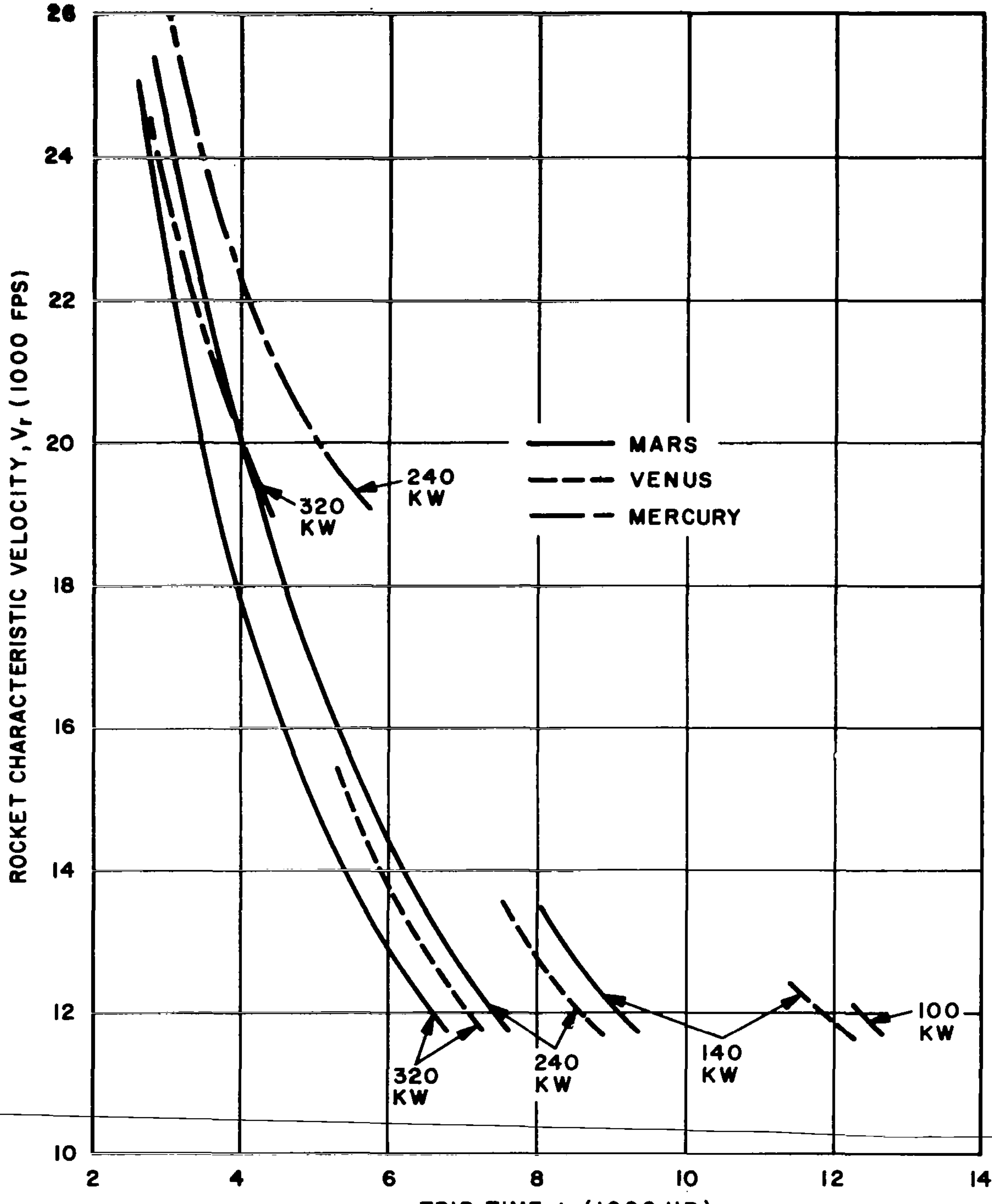

Figure 7-14. Rocket Characteristic Velocity for Minor Planet Orbiters

8.0 NOMENCLATURE

a	Low thrust acceleration, mi/hr^2
a_o	Initial low thrust acceleration, mi/hr^2
A_o	Coefficient of specific power equation, 40 kw/lb
A_1	Coefficient of specific power equation, .024 kw/lb sec
A_2	Coefficient of specific power equation, $-3.12(10)^{-7}$ kw/lb sec^2
A_o'	Coefficient of linear approximation of specific power equation, $6.9(10)^{-4}$ kw hr^2/ mi lb
A_1'	Coefficient of linear approximation of specific power equation, $1.094(10)^{-8}$ kw hr^3/ mi^2 lb
B_o	Coefficient of initial gross weight equation, lb
B_1	Coefficient of initial gross weight equation, lb sec/ft
B_2	Coefficient of initial gross weight equation, lb sec^2/ft^2
g	Sea level gravitational acceleration, 79,019 mi/hr^2
I_{sp}	Thrustor specific impulse, lb thrust/lb per sec fuel, sec, hr
J	Low acceleration propulsion parameter, mi^2/hr^3
L	Characteristic length, mi
L_o	Characteristic length extrapolated to zero trip time, mi
L_m	Minimum characteristic length, mi
P	Power rating, kw
t	Time, hr
t_h	Heliocentric trip time, hr
t_m	Time at which characteristic length minimizes, hr
t_p	Total propulsion time, hr

t_{ph}	Heliocentric propulsion time, hr
t_{pl}	Planetocentric propulsion time, hr
t_t	Total trip time, hr
T	Low thrust, lb
V_e	Earth orbital velocity, 25,000 fps
V_j	Thrustor jet velocity, mph
V_o	Initial hyperbolic excess velocity, mph, fps
V_{pl}	Planetary characteristic velocity requirement, mph
V_r	Rocket characteristic velocity, fps
w	Powerplant specific weight, lb/kw
w_t	Propellant tankage and support fraction, 9% of propellant weight
w_{th}	Specific weight of thrustor
W_o	Initial gross weight, lb
W_{pl}	Delivered mission payload, lb
μ	Final mass ratio
μ_h	Heliocentric mass ratio
μ_{pl}	Planetary mass ratio

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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